

Contents

1. Introduction	774
2. Spacecraft Wiring and Grounding	776
3. Electrical Protection Against EMI Generated by Discharge	779
4. The Deployable Solar Arrays	781
5. Spacecraft Level EMI Testing With a Spark Source	783
6. Anomalies	784
7. Conclusions and Recommendations	786
Acknowledgments	787
References	787

**6. Design, Construction and Testing of the
Communications Technology Satellite
Protection Against Spacecraft Charging**

J. Victor Gore
Communications Research Centre
Department of Communications
Shirley Bay, Station H
Ottawa, Ontario

Abstract

This paper discusses in detail the measures taken on the Communications Technology Satellite (CTS or Hermès) which provide protection against the effects of spacecraft charging. These measures include: a comprehensive grounding philosophy and implementation; provision of command and data line transmitters and receivers for transient noise immunity; and a fairly restrictive EMI specification.

Ground tests were made on materials at the Lewis Research Center (LeRC). The impact of these tests on the CTS spacecraft is described.

Hermès, launched on 17 January 1976 on a 2914 Delta vehicle, has successfully completed 10 months of operations. Anomalies observed are being assessed in relation to spacecraft charging, but no definite correlations have yet been established.

The paper ends with a list of conclusions with regard to the CTS experience and list of recommendations for future spacecraft.

1. INTRODUCTION

The Communications Technology Satellite is an experimental communications satellite, operating in geosynchronous orbit. CTS was a joint project of the Communications Research Centre, Ottawa, Canada and NASA Lewis Research Center, Cleveland, Ohio. Major components were provided by the European Space Agency.

The purpose of CTS was to demonstrate the technological and social possibilities of a high power (230 W) SHF transponder on the spacecraft used in conjunction with small low cost earth terminals. The high power demanded a three-axis stabilized configuration, large deployable sun tracking solar arrays, and thermal requirements wherein almost the entire external surface of the spacecraft was dielectric. CTS probably had more static charge accumulating area than any previous spacecraft in synchronous orbit. Figure 1 is a photograph of the flight model with the deployable arrays extended. Each deployable array blanket is 21.4 ft by 4.2 ft. The substrate is a Kapton-fiberglass laminate. The solar cells are covered by nonconducting coverglasses. On the body of the spacecraft, the dielectric areas are:

- (1) Solar cell coverglasses and fiberglass substrates cover the east and west panels.
- (2) Thermal blankets cover almost the entire forward and aft panels.
- (3) Kapton shields, not installed in this photograph, provide thermal protection for the SHF antennas.
- (4) Second surface mirrors, thermal blankets, and paints cover the north and south panels.
- (5) A thermal radiator fin extending forward from the south panel is part of a variable conductance heat pipe system that provides cooling for the SHF Transmitter Experiment Package (TEP). Both sides of the fin are covered with silvered Teflon.

The largest conducting external surfaces are the separation ring, the apogee motor nozzle, and the Bistern boom behind the deployable arrays.

For thermal and EMI reasons there are very few holes into the interior volumes. Venting was provided by a few screened apertures, loose flaps on thermal blankets that closed after ascent stage, and a thin annulus around the TEP collector. Electron penetration into the interior was thereby minimized.

The CTS project was started in 1970, and by 1974, when we fully recognized the danger presented by spacecraft charging, the engineering model tests were in progress and the flight model spacecraft and subsystems were being built. Major changes to hardware would have had an enormous impact on the project. Also,

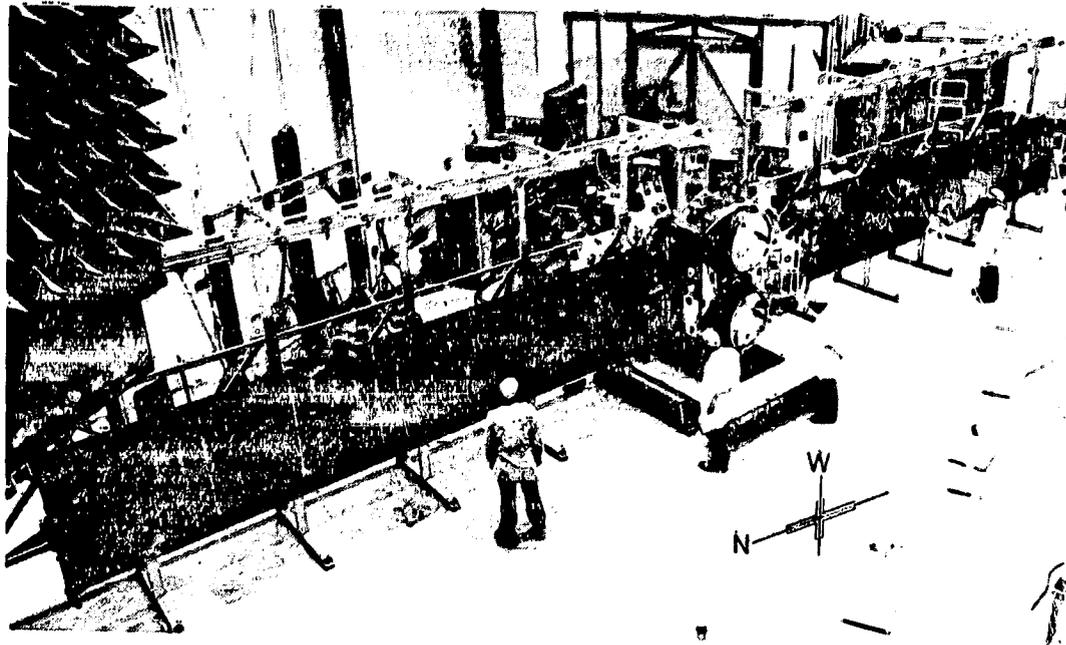


Figure 1. The Communications Technology Satellite (CTS) in the Integration Area. The deployable solar arrays extend from the north and south panels

the weight requirements were very tight. Therefore, changes were considered only if they were relatively minor and if the consequences of not making them would present a significant threat to the mission. Fortunately, during the initial design phase certain precautionary measures had been enforced that minimized the modifications necessary to provide protection against spacecraft charging effects. These measures included: stringent grounding to meet the Launch Safety Requirements as detailed in range safety manual AFETRM 127-1; stringent electrical interface design standards to ensure electrical compatibility and noise immunity; and design and test EMI requirements based on MIL-STD-461.

In late 1974, an investigation was started in Lewis Research Center, with the cooperation of CRC, to perform quickly the following tasks:

- (1) Evaluate the magnitude of the danger.
- (2) Obtain the electrical signature of discharges that would occur on external surfaces.
- (3) Prepare a test specification for spacecraft level EMI testing using a spark source. This source was to have an electrical signature consistent with the results of the second task.

John Stevens set up an electron swarm chamber¹ and ran a very rapid and prolific test program.

Primarily as a result of this work:

- (1) The grounding of thermal blankets was improved.
- (2) Teflon second surface mirrors were removed and reinstalled using conductive adhesive.
- (3) A review was made to ensure electrical connections were made between all metallic parts and surfaces to the greatest possible extent. Some improvements were made.
- (4) The EMI specification and interface protection circuits were reviewed.
- (5) A cautious EMI test using a spark source was done on the flight spacecraft.

(6) A Transient Event Counter (TEC)² was included on the spacecraft to detect high speed noise bursts on three cable harness bundles. No changes were made to subsystem components to provide extra EMI protection nor was any extra shielding added to the wiring harness.

Up to 10 months after launch we have had two major faults that could have been initiated by charge accumulation or a discharge as described in Section 6. Also, there have been several trips in a Traveling Wave Tube Assembly (TWTA) that are probably caused by noise bursts on an internal command line. No spurious commands have been experienced and the telemetry has proven to be effectively free of noise.

The remainder of this paper describes in more detail the grounding methods and the EMI protection that were used. Two special tests, one on the command belt antenna and receiver, and the second on a sample of solar array, made in the LeRC facility are briefly described. The paper ends with a list of recommendations.

2. SPACECRAFT WIRING AND GROUNDING

The CTS spacecraft uses a common electrical plane ground configuration as opposed to a single point ground scheme. For the TEP only, special ground wires are provided. The spacecraft structure primarily the north, south, forward and aft decks provide the ground return paths.

The primary power, secondary power, telemetry and commands were carried on wires grouped into four separate wire harnesses. The only spacecraft wiring that was shielded were 5 volt power lines to logic circuits, data lines between attitude control sensors and the attitude control electronics, and the reaction control system wiring.

The only changes to the flight spacecraft to provide protection against charging were made to improve the reliability of the grounding of thermal blankets and second surface mirrors, and some small metal parts.

2.1 Grounding of Individual Mounted Units

For all individually mounted units (IMU), except the TEP, the ground returns for power, telemetry, and commands are through the spacecraft structure. The primary ground connections for the IMUs are through their mounting surfaces. The requirements for IMU grounds were for less than $2.5 \text{ m}\Omega$ between the IMU and the mounting deck, and for less than $25 \text{ m}\Omega$ between any two IMUs. Secondary ground returns were provided by wires from connectors to a nearby ground lug on a mounting deck.

For the TEP separate ground wires were run from the TEP unit to the telemetry encoder, command decoder, and to the main power ground on the north deck. This arrangement kept the interface between the TEP and the spacecraft as simple as possible, and it also isolated the large current drawn by the TEP from the ground plane used by the rest of the spacecraft.

For the telemetry channels that required extra noise immunity for higher accuracy, wires were run from the ground on the telemetry encoder to differential circuits in the IMU. These wires are used for voltage reference only and do not carry return currents.

2.2 Grounding of Thermal Blankets

All thermal blankets are provided with grounding connections. Blankets with an area of less than one square foot have one ground connection, and blankets of larger area have two or more grounds.

Figure 2 shows three ground configurations, types I, V and IX. With type I and V the metallized layers, without the interlayer scrim cloth, are extended in a tab, a strip of aluminum foil 0.001 in. thick and 0.5 in. wide is woven between the layers, and the tab is then riveted with washers top and bottom. With type I a solder lug is included, and with type V the aluminum foil is extended for attachment to a deck ground. The aluminum foil provides a positive ground connection to each layer and it also provides contact over a reasonably large area. Other types from I to VIII are variations of either I or V. The type IX is a less secure ground. The aluminum foil is wrapped around the outer layers only and contact to inner layers is made by occasional contact to the rivet. No ground of type IX were used on the flight model of CTS.

2.3 Grounding of Second Surface Mirrors (SSMs) and Silvered Teflon-Electron Swarm Tests at LeRC

In the LeRC experiments, it was found that quartz SSMs produced relatively few and weak discharges. The quartz mirrors on the north and south panels were bonded with a nonconductive adhesive and were not changed. However, silvered teflon produced very frequent intense discharges and so the teflon SSMs on the

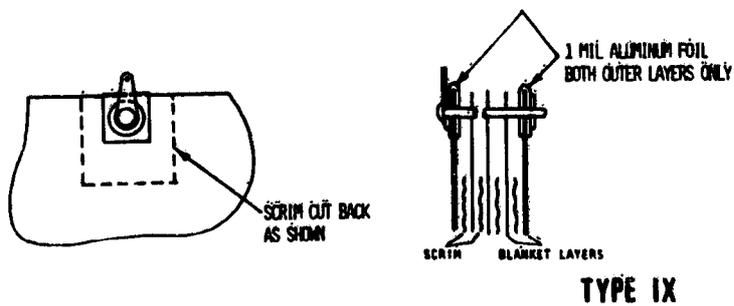
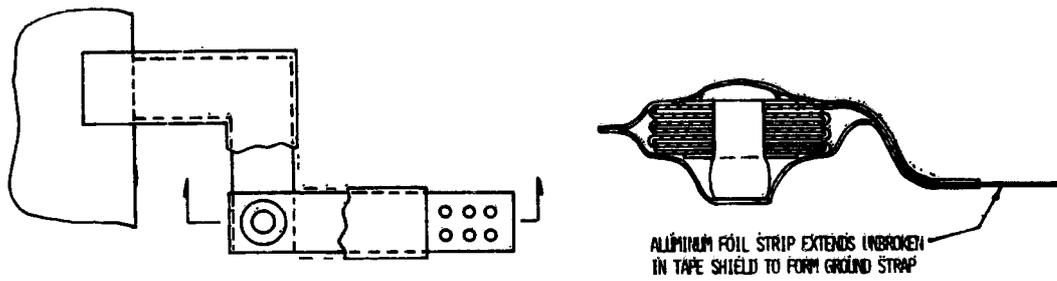
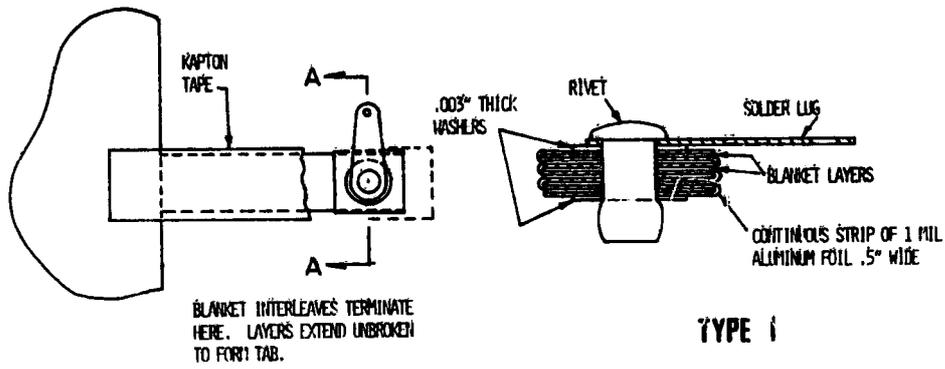


Figure 2. Standard Configurations for Thermal Blanket Grounds

north and south decks were removed and reinstalled using a conductive adhesive. The discharges on silvered teflon bonded with nonconductive adhesive not only produced electrical noise but also a considerable quantity of the silver was eroded. This indicated that during the electron irradiation the silver layer was being charged, by leakage or possibly by micro-discharges from the surface. It also meant that metal to metal arcs were occurring. In a simulated 2 year mission, test on a sample representative of the TEP radiation fin about 12 percent of the silver was eroded. This erosion would have decreased the thermal radiation from the fin by increasing the average absorptivity and by decreasing the heat conducted from the fin substrate into the teflon. The erosion could probably have been tolerated, without increasing TEP component temperatures above acceptable limits. However we felt that the EMI generated by metal to metal arcs was likely to be much greater than the discharges off the surface of the teflon. The teflon was therefore removed from most of the fin area and rebonded with a conductive adhesive.

3. ELECTRICAL PROTECTION AGAINST EMI GENERATED BY DISCHARGE

3.1. General EMI Specifications

No special measures were taken to protect units against the EMI that could be generated by discharges. However, the existing specified limits for susceptibility to conducted interference on power lines extended from 30 Hz to 40 MHz. In the range where EMI from discharges is expected, 150 kHz to 400 MHz, the limit was the 1 percent of the supply voltage or 1 volt RMS whichever was greater. Also units had to tolerate on any power rail, without malfunction, a spike of twice the nominal supply voltage or 50 volts whichever was less. The spike rise time was less than one μ sec and the duration was 10 μ sec.

3.2 Command and Data Line Interface Circuits

Command and data line transmitters and receivers were used to provide immunity from noise and fast transients on command and data lines. The characteristics of these interface circuits are listed in Table 1. Figure 3 shows typical applications for these circuits. The interface circuits were purchased to a single set of specifications and furnished to contractors as GFE items. Although not all contractors used the GFE circuits, generally alternate circuits provided similar interface characteristics.

In the spacecraft level testing with the spark source (see Section 5), the typical noise burst generated by the spark was a few volts high and about 200 nsec long. The interface circuits would protect adequately against this.

Table 1. Characteristics of the Command and Data Line Transmitters and Receivers

	COMMAND LINE TRANSMITTER	DATA LINE TRANSMITTER	COMMAND LINE TRANSMITTERS IN THE DECODER
"0" LEVEL	.25V @ -20 μ A	.25V @ -20 μ A	0 to .1V
"1" LEVEL	3.8V @ 10 mA	3.8V @ 10 mA	3.8V @ 10 mA
FALL TIME	100 μ S	1 μ S	50 μ S
RISE TIME	100 μ S	1 μ S	50 μ S
DELAY TIME "0" TO 3.2V	100 μ S	("0" TO 2.0V) 50 μ S	
DELAY TIME "1" TO .75V	100 μ S	("1" TO .6V) 50 μ S	
	COMMAND LINE RECEIVERS	DATA LINE RECEIVERS	
"0" LEVEL	.75V @ -3 μ A	.75V @ -3 μ A	
"1" LEVEL	3.2V @ 2.5 mA	3.2V @ 2.5 mA	
DELAY TIME "0" TO 2V	100 μ S	20 μ S	
DELAY TIME "1" TO .5V	100 μ S	20 μ S	
TRANSIENT REJECTION	5 VOLT PULSE 5 μ S	5 VOLT PULSE 1 μ S	
NOISE REJECTION	5V RMS @ 10 MHz	5V RMS @ 10 MHz	

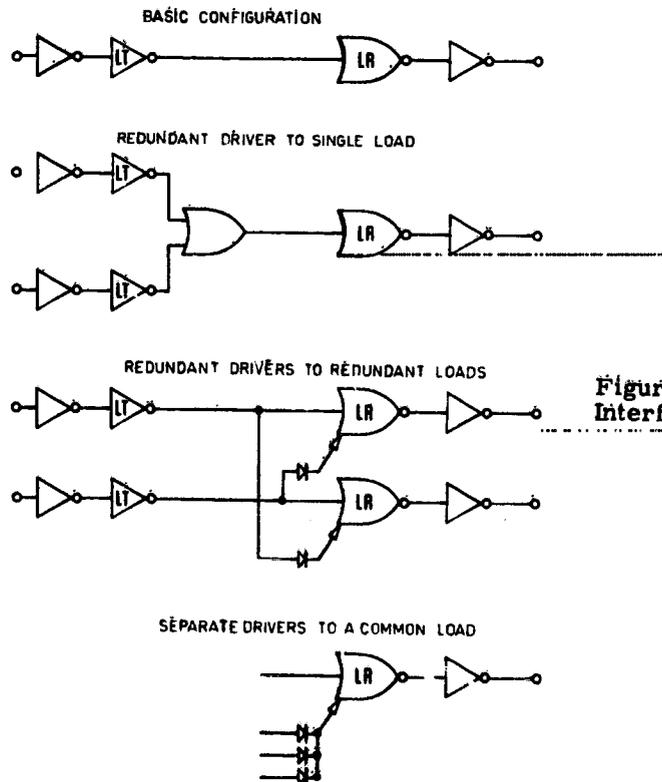


Figure 3. Typical Applications for Interface Circuits

3.3 Telemetry Interface Requirements

The requirements for analog voltages, flags and digital words that are input to the telemetry encoder are listed in Table 2. The types of input protection in the encoder are also listed. The telemetered data has proven to be effectively free of noise. Generally, analog channels show much less than one bit of noise.

3.4 Special Tests on the Command Receiver at NASA Lewis Research Center³

The telemetry/command belt antenna is a pattern of metal patches and feed lines on a fiberglass-tylon substrate. The substrate is backed with aluminum sheet. The belt antenna is completely exposed to the external particles and we were concerned about possible interference to commands or ranging, and possible damage to the thin feed lines.

Two short tests were set up in the LeRC electron swarm chamber. First, a short section of belt antenna was placed in the tank facing the electron beam. The command output from the antenna was connected into the engineering model filters and command receiver. The receiver ranging and command outputs were monitored. With a 20 keV beam discharges did occur though not visibly on the belt. The largest arcs produced a short duration 2 sec, 800 mV pulse on the receiver ranging output. No disturbance was apparent on the command output. The pulse on the ranging mode was judged to be of no consequence. For the second test, the belt antenna was placed in the tank with the metal ground plane facing the electron beam and the forward side facing a 6 in. x 8 in. sample of silvered teflon. The teflon produced frequent high amplitude arcing. No effects were observed on the receiver ranging or command outputs. After the test the belt was carefully examined; no damage was apparent on the feed lines.

4. THE DEPLOYABLE SOLAR ARRAYS

The large light weight deployable solar arrays are unique to synchronous orbit spacecraft. The array substrate is 0.001 in. thick kapton laminated with a fiberglass/polyester layer. The solar cells are carefully bonded to the substrate so as to leave the gaps between the cells free of opaque adhesive. The substrate is about 30 percent transparent to sunlight, the glass fibers will scatter some light into the areas behind the solar cells. Therefore there will be a certain amount of photoelectron emission from the rear surface, and the substrate conductivity may be increased by photoconduction effects. The arrays are nevertheless the largest area of external shadowed dielectric on the spacecraft and consequently are possibly sources of frequent discharges.

Table 2. Interface Requirements for Telemetry

ANALOG SIGNALS	
NOMINAL RANGE	0V TO 5.0V
FULL SCALE	0V TO 5.1V
ONE BIT	20 mV
SOURCE IMPEDANCE	<5 K Ω
LOAD IMPEDANCE DURING SAMPLING	1.0 M Ω
ENCODER INPUT PROTECTION BEFORE ANALOG SWITCHES	4.7 K Ω RESISTOR TO DIODE CLAMPS TO +5.6V AND TO GROUND
FILTER AFTER THE ANALOG SWITCHES-TIME CONSTANT	10 μ S
FLAGS	
FLAG	4.0V TO 5.5V
ZERO	0V TO .7V
DECISION THRESHOLD	2V
RISE TIME	1 μ S TO 10 μ S
SOURCE IMPEDANCE	<5 K Ω
ENCODER INPUT PROTECTION	75 K RESISTOR TO DIODE CLAMPS TO +5.6V AND TO GROUND
DIGITAL WORDS	
WORD "1"	4.0V TO 8.0V
ZERO	0V TO .25V
RISE TIME	1 μ S TO 10 μ S
SOURCE IMPEDANCE - "1"	<1.5 K
"0"	<2.5 K
ENCODER INPUT PROTECTION	

Tests were made on a small section of array in the LeRC chamber, with electrons incident on the rear side of the array. In a dark chamber, weak discharges occurred with visible points of arcing, primarily in the vicinity of interconnects between the cells. Measurements of the electrical output of the cells before and after the test showed that no deterioration resulted from the discharges. No changes to the arrays were considered because the flight units were completed at this time and only a catastrophic failure would have forced any modifications.

Flight data has shown no unusual electrical degradation of the arrays.

Launch ESTEC has made further tests on an array sample,⁴ with both a dark chamber and with a light source illuminating only the solar cell side of the sample. They have observed discharges with an illuminated sample at temperatures -10°C . Except for transit through eclipse and immediately after the eclipse, the lowest temperature of the flight arrays is 40°C . This test would indicate that there are possibly no discharges off the arrays in sunlight, again except for a few minutes after eclipse exit.

5. SPACECRAFT LEVEL EMI TESTING WITH A SPARK SOURCE

Based on measurements in their experiments, the LeRC group made a spark source for the spacecraft level EMI test. The source discharged a 500 pF capacitor, charged to 10 kV across a spark gap. The discharge was underdamped with a ringing frequency of 25 MHz. The spark repetition rate was about 5 pulses per second.

A series of tests were made on the flight spacecraft. These tests were cautious because we were using the flight spacecraft with no previous test experience. Also we were very uncertain as to the magnitude of the surface potentials, and discharge current's that would be experienced in space. A report by Shkarofsky and Tam⁵ indicated that the rear surface of the deployable array blanket could charge to 4.4 kV and that the corresponding energy density would be 0.27 mJ/cm^2 . The 2.5 mJ in the sparker would therefore correspond to discharging 100 square centimeters of blanket. The same report indicated that the TEP radiator fin could charge to 23 kV and the 2.5 mJ would correspond to a discharge of only 10 square centimeters. It was felt that these figures, and the test results from LeRC represented "worst case" situations; however, they were sufficiently alarming to justify the spacecraft level test.

For the test, the deployable arrays were deployed as shown in Figure 1.

5.1 Deployable Array Tests

The purpose of this test was to simulate the occurrence of arcs on the rear surface of the deployable array blanket. The procedure was for the spark source to be held about 1 m from the rear surface of the blanket and to be moved slowly towards the blanket. During the arcing, the telemetry output connection on an accelerometer mounted on the tip of the array was monitored with a high speed oscilloscope. Each spark generated a noise burst. For reasons of caution the approach towards the array was stopped when the noise burst exceeded 6 volts peak to peak. This was done at three vertical positions, first opposite the instrumentation wiring on the array centerline, second opposite the areas containing solar cells, and third opposite the power wiring along the edge of the blanket. At the centerline positions the closest approach was about 30 cm, at the power wiring the closest approach was about 3 cm (as close as the cover on the spark source allowed). During the test telemetry channels associated with instrumentation on the arrays was monitored on strip charts. After the tests a computer search of all the spacecraft channels was made. During all the sparking no telemetry data was observed that could have been attributed to the arcing.

5.2 TEP Radiator Fin Test

For this test the spark source was held close to the radiator fin, on both sides, but at some distance from sensor wiring. Selected telemetry channels associated with the TEP, particularly those temperature sensors mounted on the fin were monitored on strip charts. No change in output was observed.

5.3 Thermal Blanket Tests

In this test the spark source was held near thermal blankets on the forward deck, but for reasons of caution, not near to sensors. The computer search showed no unusual telemetry data.

Obviously a more comprehensive set of tests would have been desirable but only with previous test experience on nonflight hardware.

6. ANOMALIES

The power subsystem is divided into two independent sections: housekeeping and experiments as shown in Figure 4. The housekeeping section has three power sources, the body mounted solar array, the deployable array and batteries. All sources are connected to the unregulated bus with isolating diodes. The experiment section obtains power from the deployable arrays only. The voltage sensor on the experiment unregulated bus is on the array side of the over/under voltage switch.

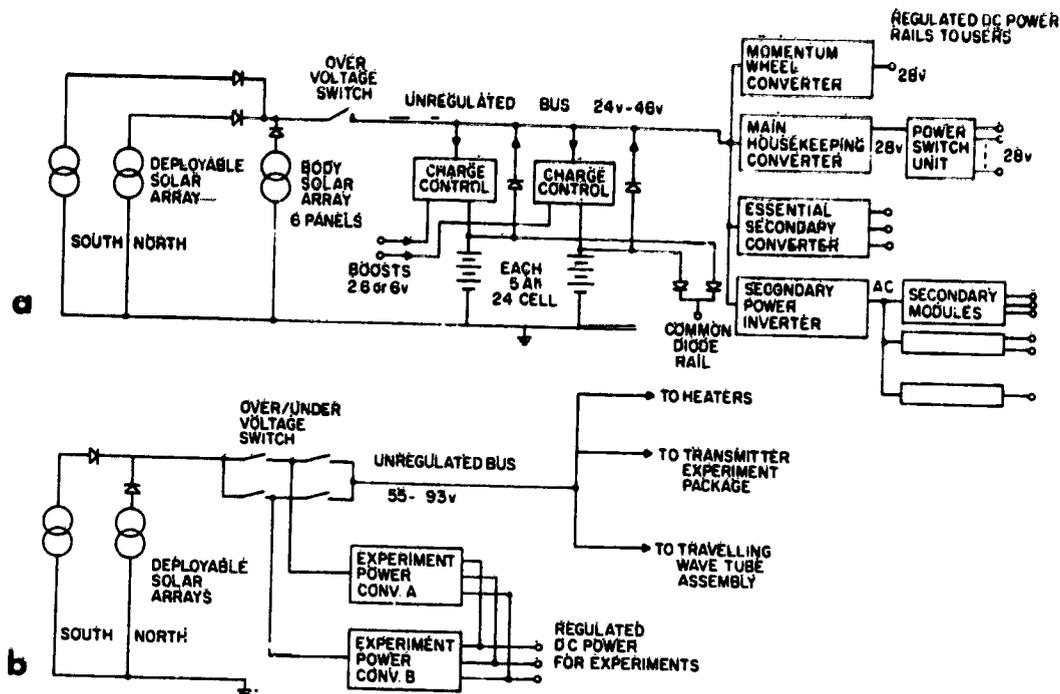


Figure 4. Power Subsystem. (a) Housekeeping section, and (b) experiment section

On 8 June 1976 at 06:43 GMT (22:59 spacecraft time) a short circuit occurred on the experiment unregulated power bus. Immediately the short occurred, the over/under voltage switch disconnected all spacecraft loads from the arrays. The short continued 24 sec then cleared itself. The most probable location for the short circuit was on a board containing the power isolating diodes. This board is mounted on the inboard pallet on the deployable solar arrays, and is completely exposed to the external environment. During the 24 sec the voltage on the experiment bus varied erratically between 19 and 42 volts, behaviour typical of an arc. Also the TEC recorded many noise bursts on the array power and instrumentation lines during the 1st, 18th, 21st, 22nd and 25th sec. These bursts were probably generated by the short circuit. The TEC had not recorded any transients in the 2 hr proceeding the fault. A moderate substorm had been observed a few hours earlier. A micrometeorite hit, charge accumulation or a small discharge resulting from charge accumulation on the diode board could have initiated the short circuit. The fault has not reoccurred.

On 28 September 1976 the housekeeping Secondary Power Inverter and all associated submodules tripped off. Unfortunately, this inverter supplied power to

the telemetry encoders so we immediately lost data. The Inverter was turned on some hours later and it has functioned normally since. A very large transient on the power bus could have caused this fault. The TEC had not registered counts in the previous hour and the magnetosphere was quiet. This anomaly is still under consideration.

Since launch there have been about 20 unexplained trip-offs on the power converter (EPC) associated with a 20 W TWTA. The EPC can get its turn off command on an internal command line. The EPC does not use a command line receiver on this line and it is known that a noise burst will result in a trip. None of the trips occurred coincident with TEC counts or with other spacecraft commanding. However, it is possible that discharges could be introducing fast noise bursts and causing the trips.

7. CONCLUSIONS AND RECOMMENDATIONS

During the first 8 months of flight operations on CTS, we have experienced:

- (1) No anomalies commands (except the 20 W TWTA trips).
- (2) Effectively noise free telemetry.
- (3) Two major anomalies that could possibly have been caused by charge accumulation or discharges.

(4) Several trips on the 20 W TWTA power converter where a command line transmitter receiver was not used.

- (5) No unusual degradation on the electrical performance of the flexible arrays.

From a spacecraft charging point of view, the important design features on CTS are:

- (1) A unified ground plane configuration was used, thus saving the weight of return wiring.

(2) Command line and data line transmitters and receivers were used to achieve immunity from fast transients.

- (3) All layers of the thermal blankets and all metal parts were grounded.

(4) Teflon second surface mirrors were bonded using conductive adhesives.

(5) Few wires were shielded, therefore the wiring harness was as light as possible.

- (6) Very little weight was added for protection against charging.

(7) A simple transient event counter was flown.

Based on the CTS experience, I would make the following recommendations:

- (1) Use command and data line interface circuits which provide protection against short, high level transients.

(2) Bond second surface mirrors including quartz ones with conductive adhesives.

(3) Ground all layers of all thermal blankets using a ground configuration of Type I or better.

(4) Ground all metal parts.

(5) Carry out a spacecraft level test on the engineering model, using a very fast spark source, to establish the electrical signature of transients on power, command, and telemetry lines. These data should then be used to specify EMI protection on flight model units. A comprehensive test should be made on the flight spacecraft.

(6) EMI specifications should include limits on emission and susceptibility on telemetry and command lines.

(7) Transient sensors should be carried. These sensors should be of more comprehensive and sophisticated design than the CTS TEC.

Acknowledgments

I would like to thank the people in NASA Lewis Research Center, System Engineering Branch, Spacecraft Technology Division under R. R. Lovell for their participation and support on the CTS charging investigation. I would also like to thank the CRC, Spacecraft Operations Group for their assistance in preparation of this paper.

References

1. Stevens, N.J., Lovell, R.R., and Gore, J.V. (1976) Spacecraft charging investigation for the CTS project, spacecraft charging by magnetospheric plasmas, A. Rosen, Editor, AIAA Progress in Astronautics and Aeronautics, 47.
2. Stevens, N.J., Lovell, R.F., and Klinec, V.W. (1976) Preliminary Report on the CTS Transient Event Counter Performance through the 1976 Spring Eclipse Season, NASA Technical Memorandum NASA TMX-73487.
3. Effects of Spacecraft Charging on the TT&C Receiving System (1975) CTS Project Test Report TR15-03-1.
4. Bogus, K.P. (1976) Investigation of a CTS solar Cell Test Patch under Simulated Geomagnetic Substorm Charging Conditions, ESTEC Internal Working Paper TEC/DB/lml/0469.

5. Shkarofsky, I. P., and Tam, S. Y. K. (1975) Investigation of the Charging of External Surfaces on CTS Synchronous Orbit and Consequent Phenomena, RCA Report No. FXC 84-1, Lab. Report No. PSP-114, MNL D-75-TR-007.